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ESSEX CORPORATION • 303 Cameron Street, Alexandria, Virginia 22314 •

TELEOPERATOR DOCKING SIMULATION

Prepared by:

Mark Kirkpatrick, Ph.D
Ronald G. Brye

ESSEX CORPORATION
303 Cameron Street
Alexandria, Virginia 22314

ESSEX CORPORATION
Huntsville Operations
11309-B South Memorial Parkway
Huntsville, Alabama 35803

Prepared for:

NATIONAL AERONAUTICS & SPACE ADMINISTRATION
Marshall Space Flight Center
Huntsville, Alabama 35812
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1.0 INTRODUCTION

Teleoperator systems are currently being considered by NASA for a wide range of space missions including orbital applications and lunar and planetary exploration. A teleoperator is defined as a man-machine system which extends and augments man's sensory and manipulative capabilities. Within the general NASA teleoperator effort, Marshall Space Flight Center has responsibility for free flying teleoperator technology development. A free flying teleoperator is a remotely controlled device which is capable of docking with and servicing a satellite in low orbit.

The teleoperator mission, as currently conceived, involves a satellite in earth orbit which requires some type of servicing. This may take the form of repair, refurbishment, or collection of data. In any case, the free flying teleoperator performs these functions under the remote control of a human operator. The teleoperator itself might be deployed from the shuttle and the operation might be situated in the shuttle or on earth. The teleoperator would contain propulsion, sensor, docking, and manipulator systems. Communication would involve deployment from the shuttle, translation to the vicinity of the satellite, final approach, and docking with or grappling of the satellite. Following these functions, the teleoperator might employ manipulator arms to accomplish the servicing operation on-site or might return and place the satellite in the cargo bay.

The current report describes the first phase of a simulation study of the translation, station-keeping and final approach segments of the teleoperator

mission in accordance with teleoperator technology development plans being implemented at MSFC. The present study represents one part of an effort to identify the teleoperator visual system design parameters which influence operator performance and to determine human factors design requirements for free flying teleoperators.

General Approach

The simulation approach employed a six degree-of-freedom motion generator to impart apparent motion to a scaled satellite model. The operator viewed the target satellite via closed circuit television and attempted to close the range to the satellite via translation and attitude controls. The control commands from the operator's station were sensed by a hybrid computer system which solved a sixteen thruster propulsion system math model using assumed vehicle mass, thrust, and dynamic parameters. The resulting position and attitude parameters were then used to control the target motion generator. The hardware components of the simulation system are shown in Figure 1.

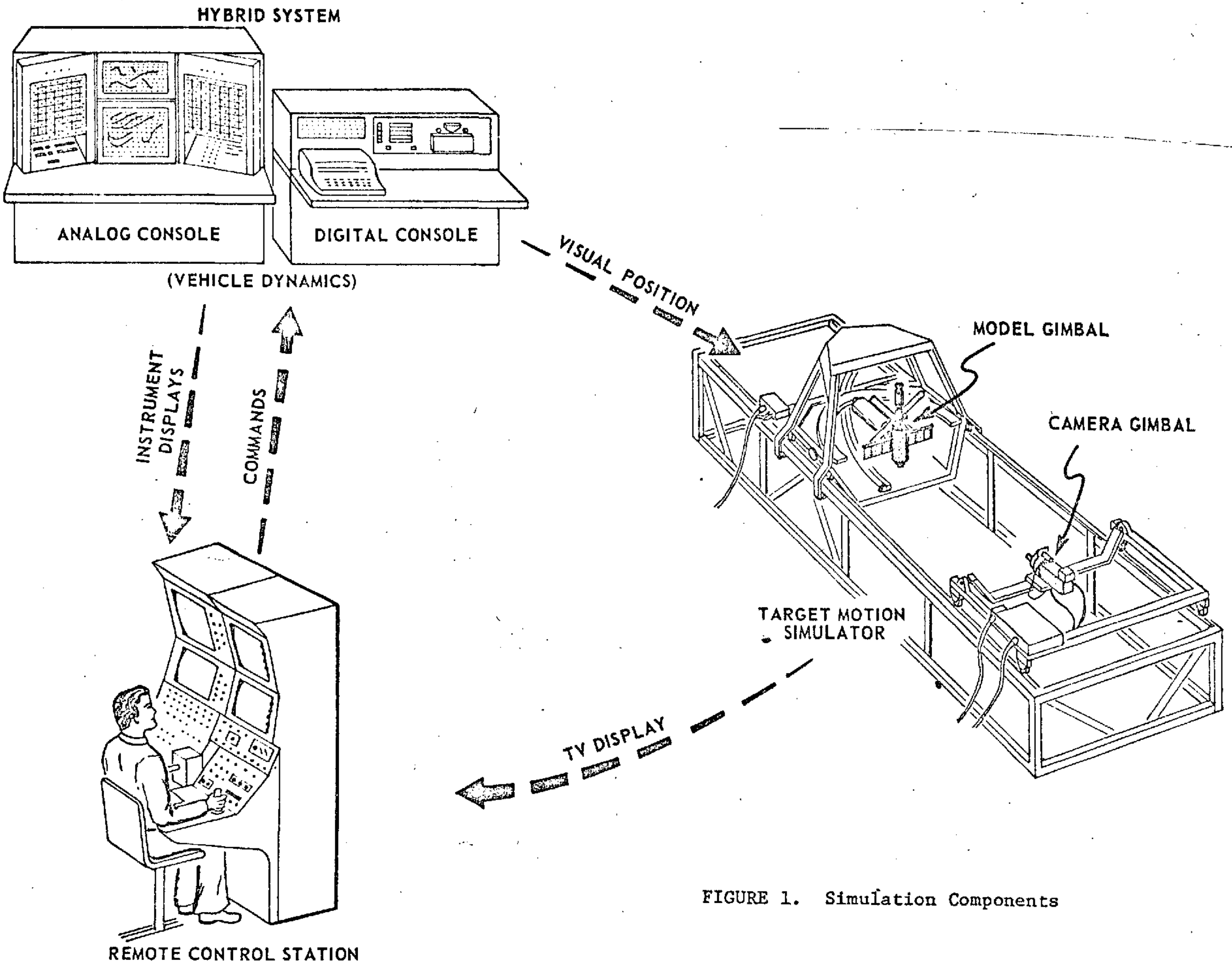


FIGURE 1. Simulation Components

2.0 SIMULATION TECHNIQUE

The subsystems which were integrated to produce the docking simulation included the following:

- Target Motion System
- Operator's Station
- Analog-hybrid Computer System

2.1 Target Motion System

The target motion system (TMS) used a series of servo-controlled gimbal systems to produce relative position and motion between the satellite model and the TV camera. The camera was mounted in a two axis system providing pitch and yaw. The satellite model was placed in a three-axis system having pitch, roll, and yaw capability. Translation in Y and Z were simulated by commanding simultaneous opposite sign yaw and pitch excursions of both the satellite model and the TV camera.

The satellite model was painted with fluorescent paint and the TMG apparatus was painted a non-reflective black. During testing the target was illuminated with ultraviolet lamps which, when viewed against the black non-textured background, provided a realistic image on the monitor.

2.2 Operator's Station

The operator's station was an Air Force F-86 function mockup. The primary display was an 18 inch (diagonal) monitor displaying the satellite view. The monitor was equipped with a concentric ring reticle similar in design to that employed in the Visual Test Laboratory. In addition to the TV display, the console contained meters displaying range, attitude, range rate, and attitude rates.

Translation control of the teleoperator was provided by a three degree-of-freedom discrete-signal band controller. Attitude control was via a three degree-of-freedom rate proportional hand controller. When the attitude controller was in the null position, an automatic attitude control system (ACS) maintained constant attitude within a specificd tolerance or deadband.

2.3 Analog-Hybrid Computer System

The computer system accepted the operator's control inputs and solved a mathematical model of satellite/teleoperator dynamics and relative motion. The propulsion system assumed for the simulation was the baseline system proposed by Bell Aerospace (Fornoff, et al, 1972). The analog math model employed was the sixteen-thruster model derived by NASA MSFC personnel to support docking studies performed for the Apollo Program. A Description of the assumed parameter values and equations used for the simulation is being prepared by MSFC Computations Lab personnel and will be published in February 1974.

2.4 Experiment Procedure

The general testing procedure was conducted in four discrete phases as follows:

Phase 1: Translate from initial starting position (approx. 70 meters in X and various displacements along the Y and Z axes and attitude conditions) to ten meters and null the closing velocity. Report to experimenter when this accomplished.

Phase 2: Station-keep at this position, inspect the satellite, estimate satellite nutation conditions and report to experimenter and when ready enable the docking probe by depressing a switch in the cockpit (there was no visible probe deployed).

Phase 3: Track the satellite attach point. Align the teleoperator X-axis with respect to the satellite X-axis and command a closing velocity. The subject reported to the experimenter when he began this final closure.

Phase 4: Final approach and dock. Subject closed to satellite attach point while continually modifying his attitude, position, and closing velocity to simulate a dock. The target satellite and camera did not make contact for a successful dock but when range was approx. 0.61 meter (simulated), the computer stopped the system. This signalled the end of a trial. During this phase, if a negative command of five meters in the X-axis was commanded, an abort was recorded.

All subjects were instructed as to the operating controls, sequence of four phases, and each allowed practice trials to gain familiarity with the system and sequence of operation. Each trial began with the satellite displaced at a random position and attitude subject to the constraint that the model was always within the TV field of view. The satellite was in one

of two modes; either stationary or nutating at various angles and rates of revolution. Controls were then "released" to the subject and initiation of Phase 1 began. When the subject reached the 10-meter range and nulled velocity he reported "Ready". The experimenter pressed a switch which allowed the dependent measures to be recorded on the digital system. Phase 2 then began and when the subject reported satellite conditions the experimenter instructed the subject to throw his "Probe Enable" switch in the cockpit which signalled the end of that phase. Again the dependent measures were recorded by the digital system. Bhase 3 was conducted in a similar manner; the subject reported "Ready" and the experimenter depressed a switch thereby recording the dependent measures on the digital system. Phase 4 was then initiated and ended when the simulated range was approximately 0.61 meter. Again the dependent measures were recorded on the digital system. During this phase, if the subject commanded a positive Δ range of five meters, the experimenter considered this as an abort and recorded the fact.

Four subjects were selected and used throughout the test. Subject selection criteria were pilot experience, education, and good vision. Three subjects were licensed pilots with greater than 300 hours of flying time. The fourth subject was not licensed but was under consideration by NASA for astronaut training so it was decided to compare his data with the veteran pilots. All subjects were degreed in technical fields and all had good vision as indicated by recent flight physicals. The subjects ranged in age from 28 to 42. A secondary subject selection criteron was available time of subject since a group of tests consumed about half a day and the

tests could only be scheduled during working hours; therefore, each subject was required to be free one-half day per week.

2.5 Dependent Measures

Three primary dependent measures were employed in the initial study reported here. As noted in the section on the experiment procedures, the approach and docking mission segment was divided into four phases according to functional requirements. The completion of each phase was indicated to the computer by a discrete signal from either the subject or the experimenter. Consequently, the dependent measures could be scored for each phase and these single phase data could be combined to yield a corresponding measure for the total mission.

For each of the four phases, two measures were recorded - phase completion time and propellant consumption. The latter was expressed as a percentage of the initial propellant available. A third measure - number of docking aborts - was recorded only during Phase 4.

3.0 BASE LINE EXPERIMENT

3.1 Independent Variables

The independent variables planned for study in the docking simulator include:

- Satellite type - Bio Research Module or Large Space Telescope
- Satellite nutation angle
- Satellite nutation rate
- Initial starting position and orientation
- TV aids - present or absent
- Range, range rate displays - present or absent
- Attitude control system - deadband

This report covers an initial experiment carried out using the Large Space Telescope with both TV aids (concentric ring reticle) and range/range rate displays present. Initial starting position and attitude were randomized with initial range maintained at 70 meters. The nutation angle and rate parameter values studied are shown in Table 1.

TABLE 1. Combinations of Nutation Angle and Rate

Nutation Angle (Degrees)	Nutation Rate (R.P.M.) 0	2	4	5	10
0	+*	-	-	-	-
2	-	+	+	+	+
4	-	+	+	+	+
5	-	+	+	+	-
10	-	+	+	-	-

*denotes treatment combinations included in the experiment

The three parameter combinations in the lower right corner of Table 1 were originally included in the study but were found to preclude successful docking in a majority of cases. Under these three conditions each subject attempted three runs and no successful final approach was obtained in any of these runs. Consequently, these conditions were dropped from the study. Each of four subjects made at least one run under each parameter set denoted by a (+) in Table 1.

3.2 Procedure

The satellite was presented to the subject via his TV monitor under a preset initial displacement, satellite nutation, and T/O ACS dead-band. The subject was then instructed to proceed with translating towards the satellite until he was ten meters from the face of the satellite, null his forward rate, by observing his range and range rate displays, and report to the experimenter. This would complete Phase 1 of that trial and the data were recorded. While at this ten meter range the subject attempted to match the satellite nutation (if any) by commanding translation and attitude. When this was done the subject threw his "Probe Enable Switch" inside the cockpit which ended Phase 2. The data were recorded at that time. The next step was for the subject to hold the T/O in that attitude and position and begin his final approach. When the subject began his final approach he reported to the experimenter, who signaled the end of Phase 3 by depressing a switch thereby allowing the data to be recorded. The subject then maintained this final approach until a yellow lamp came on in the cockpit signaling him he had docked. The data at this point were recorded via the digital computer and this completed Phase 4 of the trial and also the end of that trial. The TMS was reset with new parameters for the next trial and the same procedure repeated. Each subject completed all four phases of a trial before it was considered complete and a new trial begun.

3.3 Results

Mean elapsed time and mean per cent propellant expended were analyzed as functions of nutation angle and rate. The effect of nutation angle on elapsed time is shown in Figure 2, which depicts total mission elapsed time and Phase 4 elapsed time. Linear functions have been fitted to the data by the method of least squares. It may be seen that nutation angle exerts a pronounced influence on both elapsed time measures. The similarity of the estimated slope parameters suggests that the increase in mission time due to nutation angle is primarily due to increased time required in Phase 4.

Figure 3 displays mean elapsed time as a function of nutation rate. Nutation rate effects appear less consistent than those of nutation angle, the data points being more widely dispersed about the line of best fit. As in the case of nutation angle, the nutation rate effect is primarily localized in Phase 4.

Mean per cent of initial propellant consumed is shown as a function of nutation angle in Figure 4 . Here, the percentage of propellant consumed during approach ranges from about 3% for a stable satellite to more than 9% for 10° nutation. Since fuel expenditure and elapsed time would be expected to be correlated, it is not surprising that the former measure is strongly influenced by nutation angle. The slope estimates for total mission and Phase 4 propellant consumption agree closely suggesting that the effect is confined to Phase 4.

Figure 5 shows propellant consumption as a function of nutation rate. While the fitted functions show an increase in fuel usage with nutation rate, the data points are widely scattered about this trend.

In comparing the data, the influence of nutation angle on both total mission time and total propellant consumption are considerable and outweigh

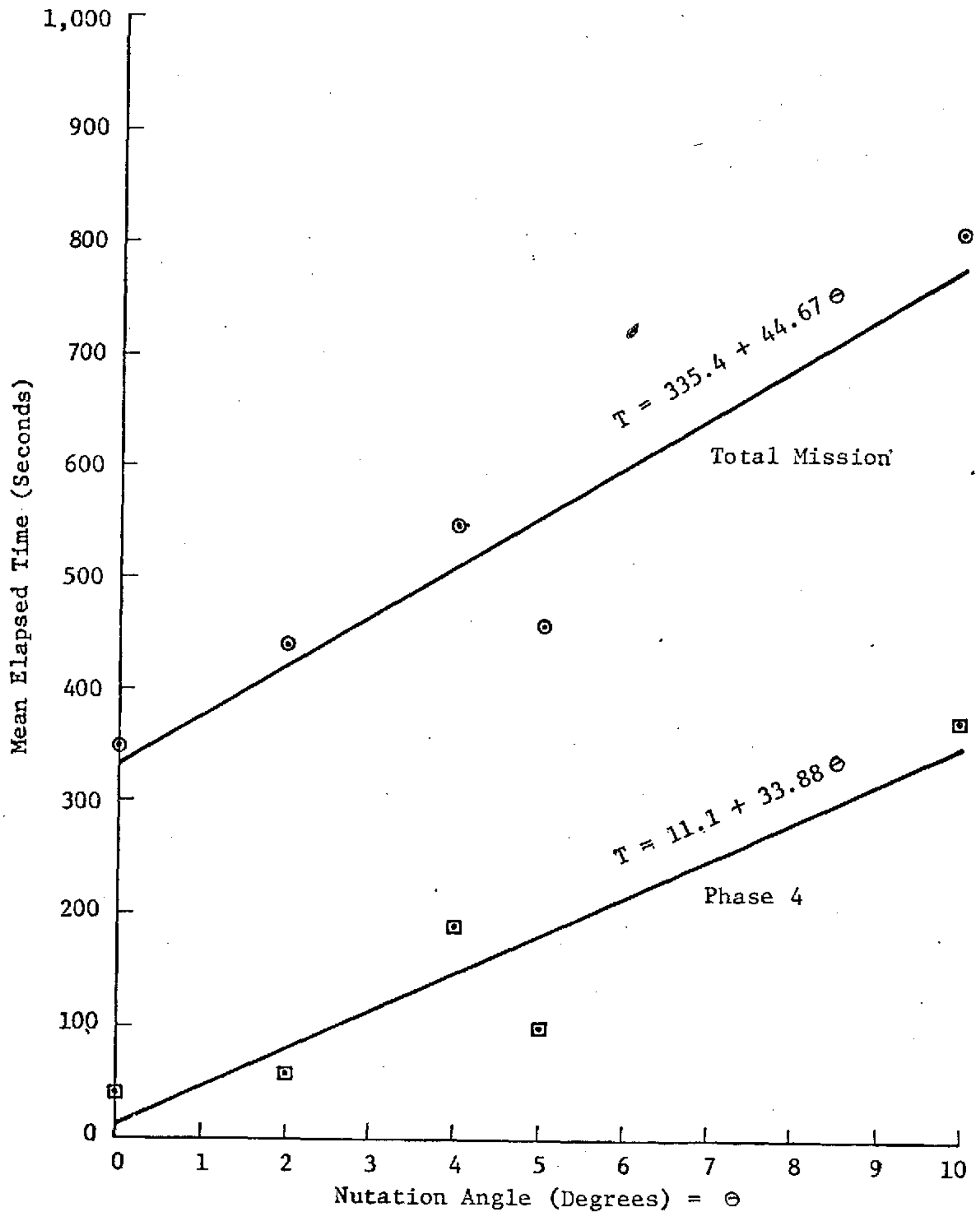


FIGURE 2. Mean Elapsed Time as a Function of Nutation Angle

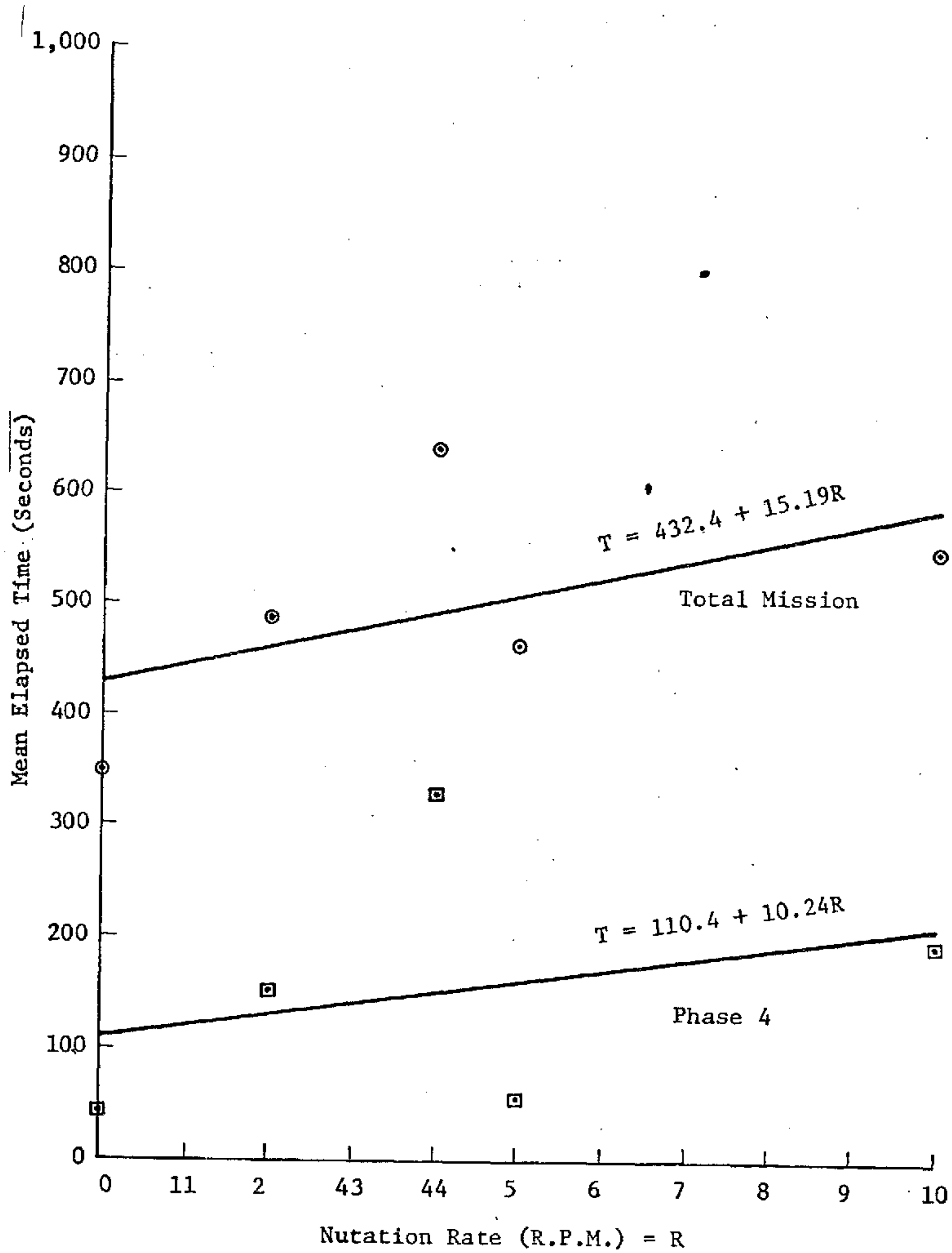


FIGURE 3. Mean Elapsed Time as a Function of Nutation Angle

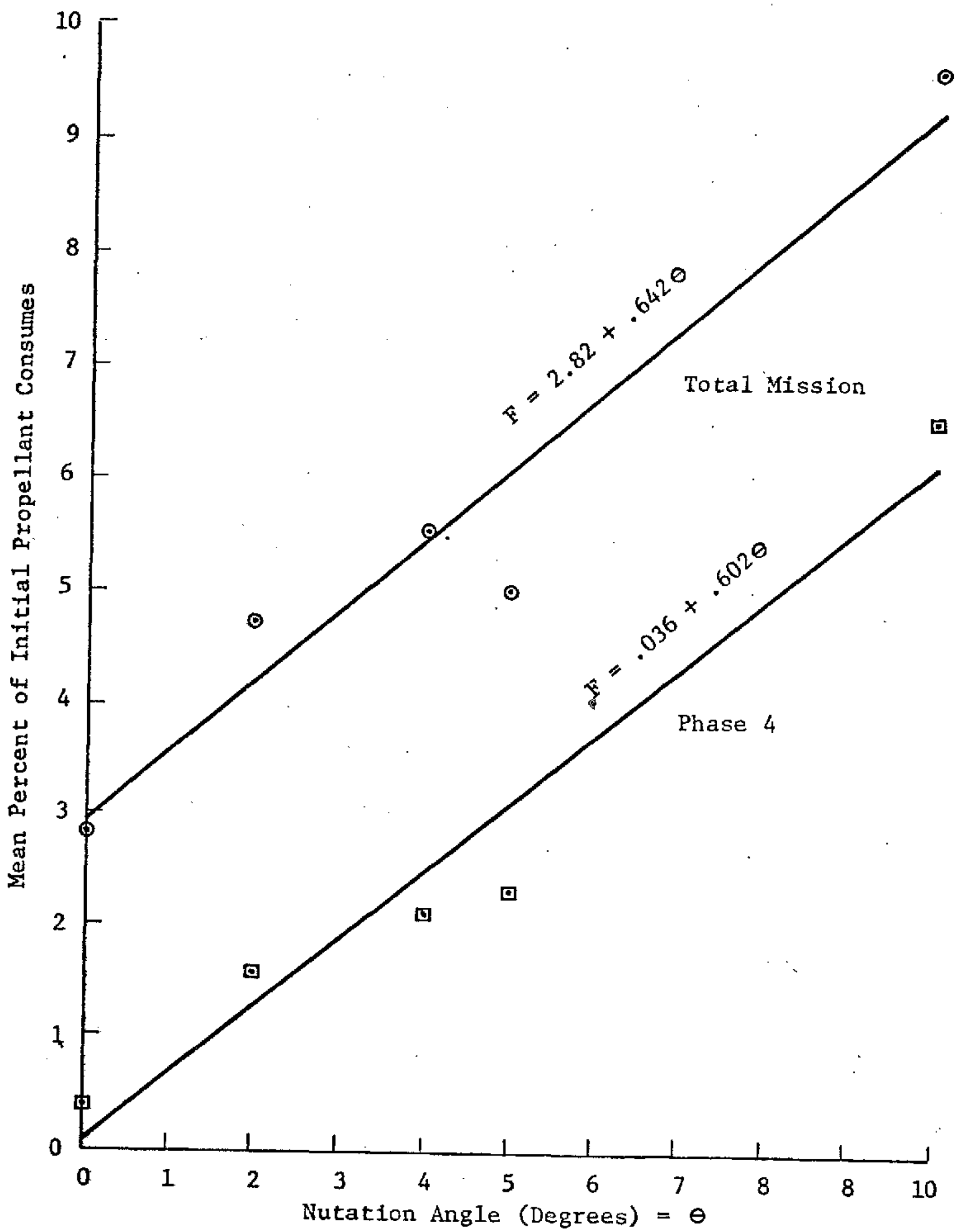


FIGURE 4. Mean Elapsed Time as a Function of Nutation Angle

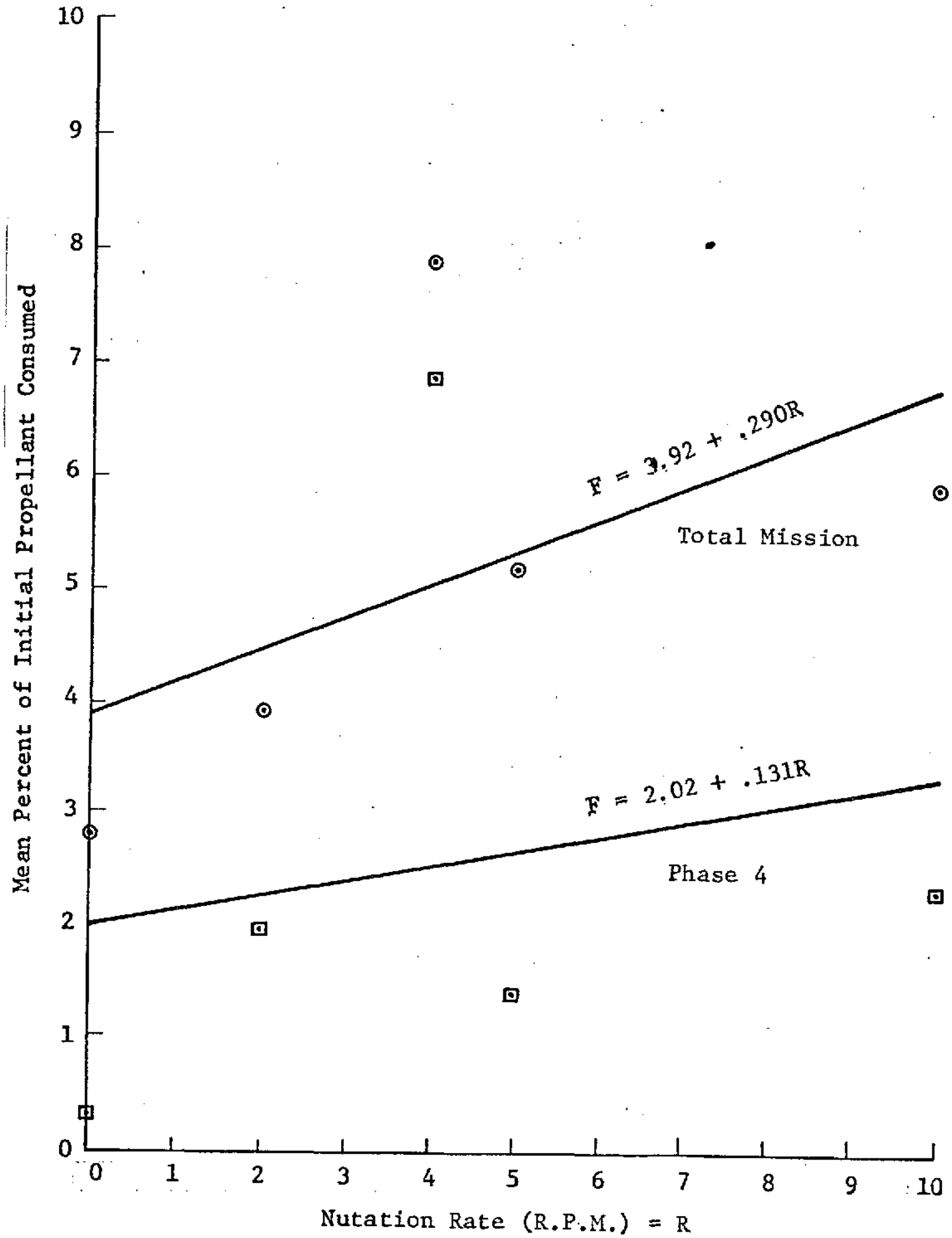


FIGURE 5. Mean Percent of Initial Propellant Consumed as a Function of Nutation Rate

the effects of nutation rate. The per cent of dependent measure variance accounted for by each independent variable is shown in Table 2 .

Dependent Measure	Independent Variable: Nutation Angle	Independent Variable: Nutation Rate
Mean Total Mission Elapsed Time	90.40	28.77
Mean Total Mission Per Cent Propellant Consumption	93.78	31.36

Table 2 - PER CENT OF DEPENDENT MEASURE VARIANCE ACCOUNTED FOR BY INDEPENDENT VARIABLES

It may be seen that both dependent measures are considerably more sensitive to nutation angle than to nutation rate within the range of these parameters studied here. It should be noted that addition across rows in Table is not valid since the experimental design was not orthogonal.

Debriefings were held following experimental runs. All subjects reported difficulty in maintaining satellite/teleoperator alignment for the higher nutation angles and rates. As the angle of nutation and the rate of revolution increased past the range where the T/O control dynamics could match the satellite dynamics the subject resorted to what might be called the "ambush technique." This is a method where the T/O was stationary in a certain quadrant and the subject would wait for the satellite to sweep by. When he felt the timing was right he would command a decrease in range relative to the velocity of the satellite revolution rate, and commit himself to a hard dock. If his timing was off he would immediately command a positive rate to a safer distance

and try again. This is how a majority of the aborts came about. When the subject did abort he usually lost his docking position and thereby consumed time and fuel in reestablishing his straight-in docking position.

Debriefings of the subjects after the "ambush technique" revealed they felt that a good dock was impossible so rather than consume time and fuel by attempting a straight-in dock they would settle for a "near proximity" dock. One question this highlights is that if the T/O loses visual contact with the satellite or the T/O overshoots the satellite, how does the controller relocate the satellite? Perhaps a wide-angle lens can be used to reestablish visual contact then switch to a narrower lens after contact.

All subjects reported difficulties with the size and displacement of the controls. As range decreases, constant course corrections are necessary to maintain alignment with the satellite. With rapid rates of discrete control movements, the relatively large controllers used caused difficulty due to the requirement for high frequency high amplitude movements. Possibly a variable control/display ratio should be incorporated into the system.

4.0 PLANNED STUDIES

Current planning calls for additional studies of satellite approach using only optical ranging methods. In the study reported here, range and range rate displays were available to the operator. In the immediate future, tests of performance without these displays will be conducted. Eventually, dynamic reticle and computer generated TV aids will be incorporated.

The conduct of the study of effects of removal of range and range rate displays is straightforward. The base line study will be replicated with the displays disconnected. Further development of dependent measures will be carried out. The measures collected in the base line study are <u>figures of merit</u> which do not reveal details of performance, although they are important system performance criteria. A more comprehensive set of performance measures would be derived from errors in attitude and translation control. During the approach to a satellite, translation errors are nulled if the body axes of the teleoperator and the satellite are aligned. This condition will be met if Y, Z, pitch, and roll for the vehicles are equal to required values. Consequently, four errors are possible - one for each degree of freedom. If W_T and W_S represent a particular axis value for teleoperator and satellite respectively, then:

$$E = W_T - W_S$$

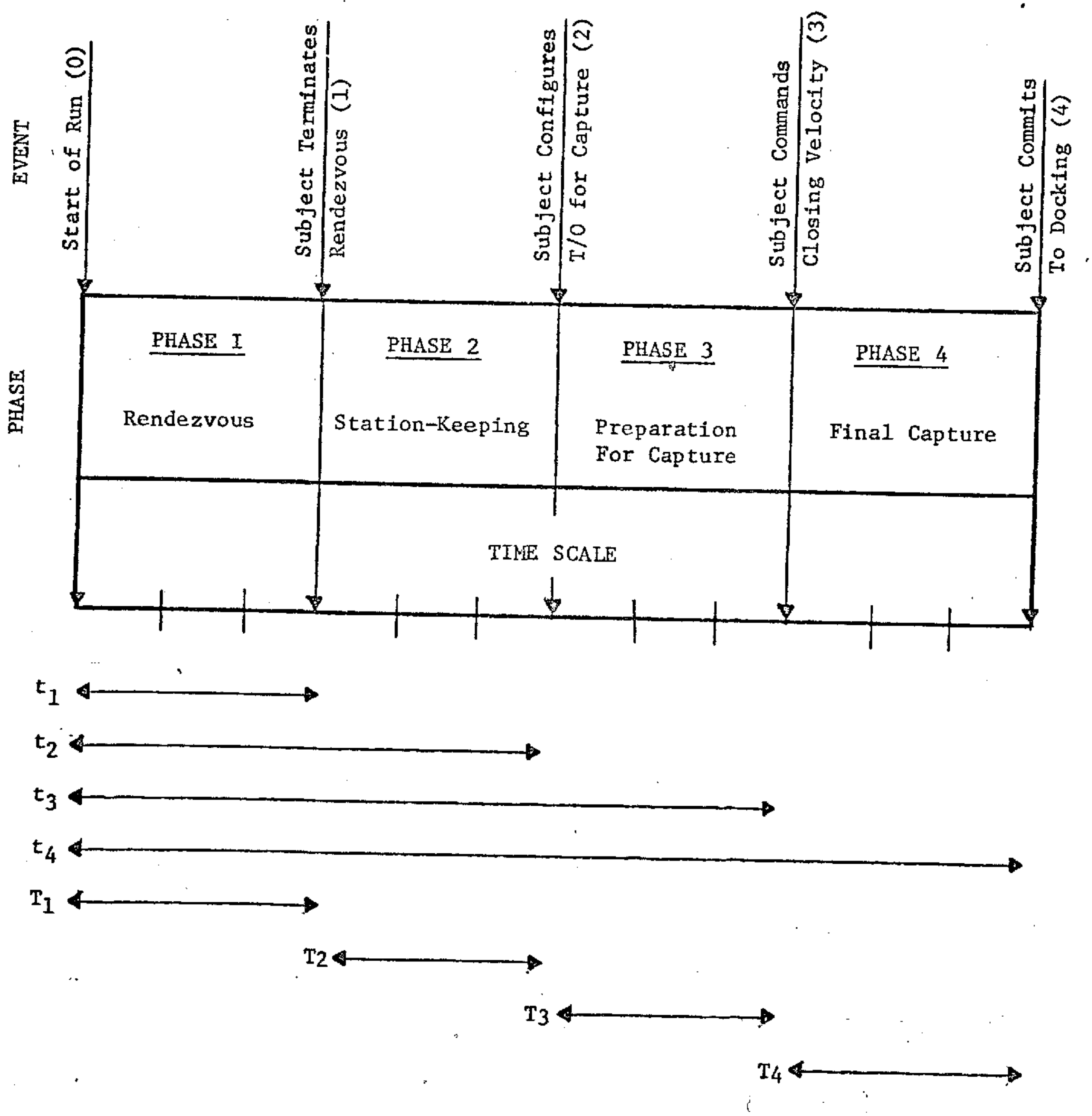
is an instantaneous error measure. The squared integrated error:

$$SS_E = \frac{1}{T}\int_0^T (W_T - W_S)^2 \, dt$$

then yields a measure of both constant and variable error. To measure the

FIGURE 6. Definition of Mission Phases

PHASE	DESCRIPTION	TERMINATING EVENT
1	Rendezvous	1 Null Approach Rate
2	Station-Keeping	2 Configure for Capture
3	Preparation for Capture	3 Command Approach Rate
4	Final Capture	4 Commit to Dock



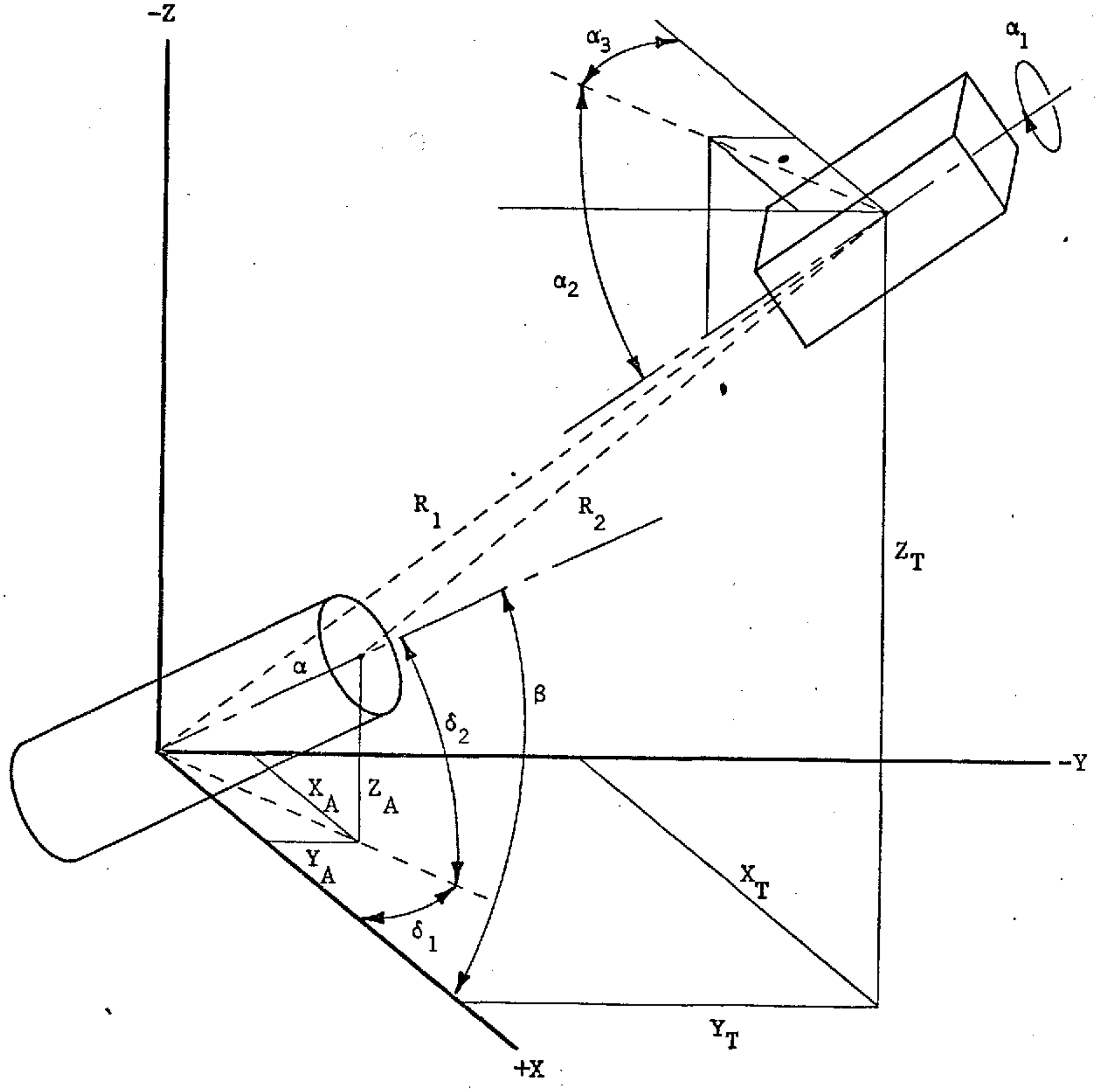


FIGURE 7. Satellite-Teleoperator Geometry

variable error alone, the mean error may be subtracted by:

$$MS_E = SS_E - \left[\int_0^T Edt^{\,2} \right]$$

The mean square errors for the four alignment degrees of freedom would permit analysis of operator performance by degrees of freedom and would allow attitude and translation performance to be studied separately.

Since instantaneous translation values would permit calculation of range and range rate, range errors during station-keeping and ability of the operator to follow a planned range/range rate profile would also be amenable to quantification. The dependent measures will be calculated by phases as shown in Figure 6 . The geometry is depicted in Figure 7 .

4.1 Definitions

a	=	distance from satellite C.G. to attach point
X_A, Y_A, Z_A	=	attach point coordinates
δ_2	=	satellite pitch
δ_3	=	satellite yaw
X_T, Y_T, Z_T	=	teleoperator C.G. coordinates
α_1	=	teleoperator roll
α_2	=	teleoperator pitch
α_3	=	teleoperator yaw
R_1	=	range from satellite C.G. to teleoperator C.G.
R_2	=	range from attach point to teleoperator C.G.
β	=	nutation angle

A nutating satellite viewed along the X axis will produce an attach point locus which is a circle normal to the X axis and of radius A where A is:

$$A = a \sin \beta$$

If $\dot{\Theta}$ is the nutation rate and:

$$\Theta(t) = \Theta(0) + \int_{o}^{t} \dot{\Theta} dt$$

Then

$$Z_A = A \sin \Theta$$

$$Y_A = A \cos \Theta$$

$$X_A = a \cos \beta \qquad \text{(constant)}$$

For the satellite body axis to pass through the teleoperator center of gravity, the latter must be located on a circle normal to the X axis. This circle must be of radius

$$B = X_T \tan \beta$$

Consequently, translational error measures E_Y and E_Z are given by:

$$E_Y = Y_T - X_T \tan \beta \cos \Theta$$

$$E_Z = Z_T - X_T \tan \beta \sin \Theta$$

Errors in attitude E_2 and E_3 are then simply

$$E_2 = \alpha_2 - \delta_2$$

$$E_3 = \alpha_3 - \delta_3$$

The following dependent measures are then recorded for the ith phase:

- Phase elapsed time $\quad T_i = t_i - t_{i-1}$
- Fuel consumption
- Terminal range $\quad R_i$

- Terminal range rate $\dot{R}_i$
- Terminal teleoperator position X_{Ti}, Y_{Ti}, Z_{Ti}
- Terminal teleoperator translation rates $\dot{X}_{Ti}, \dot{Y}_{Ti}, \dot{Z}_{Ti}$
- Terminal teleoperator attitude $\alpha_{1i}, \alpha_{2i}, \alpha_{3i}$
- Terminal teleoperator attitude rates $\dot{\alpha}_{1i}, \dot{\alpha}_{2i}, \dot{\alpha}_{3i}$
- Terminal attach point position Y_{Ai}, Z_{Ai}
- Terminal satellite attitudes $\delta_{2_i}, \delta_{3_i}$

- Mean Y error $$M_i\,(E_Y) = \frac{1}{T_i}\int_{t_{i-1}}^{t_i} E_Y\,dt$$

- Mean Z error $$M_i\,(E_Z) = \frac{1}{T_i}\int_{t_{i-1}}^{t_i} E_Z\,dt$$

- Squared Integrated Y error $$SS_i(E_Y) = \frac{1}{T_i}\int_{t_{i-1}}^{t_i} E_Y{}^2\,dt$$

- Squared Integrated Z error $$SS_i(E_Z) = \frac{1}{T_i}\int_{t_{i-1}}^{t_i} E_Z{}^2\,dt$$

- Mean pitch error $$M_i(E_2) = \frac{1}{T_i} \int_{t_{i-1}}^{t_i} E_2 \, dt$$

- Mean yaw error $$M_i(E_3) = \frac{1}{T_i} \int_{t_{i-1}}^{t_i} E_3 \, dt$$

- Squared integrated pitch error $$SS_i(E_2) = \frac{1}{T_i} \int_{t_{i-1}}^{t_i} E_2^2 \, dt$$

- Squared integrated yaw error $$SS_i(E_3) = \frac{1}{T_i} \int_{t_{i-1}}^{t_i} E_3^2 \, dt$$

These dependent measures will be collected during future docking simulation studies. Some of the applications of these measures to studies of man/system performance are discussed below.

4.2 Time Required for Phase Completion (T_i)

The time required for phase completion is an obvious figure of merit for system/operator performance. Presumably the longer the time required, the greater the difficulty of tasks associated with a particular phase. In addition, studying phase completion time as a function of the independent variables employed will permit detection of differential effects of these

variables on different tasks. For example, attitude control system effects would be expected during the final approach to a greater degree than during initial translation. Furthermore, completion time data will be required for time line planning and workload analysis. If task completion were time constrained during a mission, such data could be used to analyze the probability of task completion in connection with reliability analyses. For a task having completion times distributed with mean μ and standard deviation σ, (in seconds) and a required completion time of T seconds, the probability of success would be approximated by

$$P = \int_{-\infty}^{T} \left[\sigma \quad 2\pi\right]^{-\frac{1}{2}} \quad \exp \left[\frac{-(t-\mu)^2}{2\sigma^2}\right] dt$$

provided that μ were on the order of several minutes and that $\mu \geq 3\sigma$.

Collection of completion time data for the four phases will be carried out separately. These times could then be summed to permit analysis of total docking/grappling time.

4.3 Fuel Consumed During Phases (F_i)

The considerations which were stated in connection with phase completion time also apply to fuel consumption. This measure serves as a performance figure of merit - particularly since errors in aligning the teleoperator and satellite body axes will require correction which will be reflected in increased fuel expenditure. Data on distributions of fuel required will also be useful in determining system design requirements. Fuel expended will be measured separately during each phase.

4.4 Terminal Position, Attitudes, and Rates

At the end of each phase, the instantaneous coordinates of the teleoperator

C.G. (X_T, Y_T, Z_T) relative to the satellite C.G., the teleoperator attitude (α_1, α_2, α_3), the C.G. to C.G. range (R_1) and the corresponding rates ($\dot{X}_T$, $\dot{Y}_T$, $\dot{Z}_T$, $\dot{\alpha}_1$, $\dot{\alpha}_2$, $\dot{\alpha}_3$, $\dot{R}_1$) will be measured and recorded. These variables will have nominal values at various points in the mission. For example, stand-off and satellite inspection should be accomplished with R_1 at a planned value, all rates nulled, and position and attitude at planned values or within tolerances. The operator's ability to meet these criteria as a function of display/control and satellite motion characteristics will thus be determined and will provide design data.

The position, attitude, and rate data recorded at the end of Phase 4 are of particular interest. Due to the nature of the simulation, the actual teleoperator/satellite contact will not be made. The approach will be terminated when a simulated distance less than a meter remains. At this point the operator will be committed to docking. The final position, attitude, and rate data may then be used to project forces and loads applied to the docking probe and adaptor hardware at impact. These data should thus prove useful in the design of both teleoperator and satellites.

4.5 Number of Docking Aborts During Phase 4

In the latter portion of the final approach, the operator will decide whether the approach is satisfactory. If so, he will then commit to docking and the simulation will terminate shortly thereafter. Slight departures from a nominal approach may be corrected via appropriate control actions. If the approach is grossly off-nominal, however, the operator may elect to abort the approach, command a positive range rate, re-acquire attach point tracking, and make another approach. The number of such abort maneuvers will be recorded and will serve as a figure of merit for performance during the final approach.

4.6 Tracking Error Statistics

During Phases 3 and 4, the operator will attempt to keep the teleoperator body axis aligned with that of the satellite. At any point in time, the instantaneous position and attitude data Y_T, Z_T, σ_2 and σ_3 yield measures of error. Whenever Y_T and/or Z_T are not 0, the teleoperator is displaced from the satellite. When σ_2 and/or σ_3 are not 0, there is misalignment. To summarize these errors over the duration of Phases 3 and 4, summary statistics - the mean and variance of each error measure will be recorded. For any measure W_k, the mean during Phase i is:

$$M_i(W_k) = \frac{1}{T_i} \int_{t_{i-1}}^{t_i} \dot{W}_k \, dt$$

The mean error measure reflects primarily bias or constant error. If error "averages" zero, the mean will be zero. The variance yields a measure of variable error being the square of R.M.S. error. Then $SS_i(W_k)$ is the variance of W_k around zero. It reflects both constant and variable error. $V_i(W_k)$ measures variable error around the mean error. If: $M_i(W_k) = 0$

$$V_i(W_k) = S_i(W_k)$$

These tracking error statistics will be recorded during Phases 3 and 4. They will provide information on task difficulty as influenced by the independent variables.

4.7 Range Rate Error Statistics

During Phase 4, a nominal R-$\dot{R}$ profile will be in force. This will specify $\dot{R}$ for each value of R during the final approach. The $\dot{R}$ value obtained may be

compared with this nominal value and a measure of $\dot{R}$ error will thus be available. The mean, variance, and integrated square of this measure will also be recorded during Phase 4.